NASA TECHNICAL NOTE

NASA TN D-4699

ANALYSIS OF SEVERAL METHANE-FUELED ENGINE CYCLES FOR MACH 3.0 FLIGHT

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • JULY 1968

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ABSTRACT

Several means of using methane's heat sink capacity for turbine cooling and for other purposes to improve engine specific impulse, specific thrust, and thrust per unit compressor frontal area were analyzed. Preheating of the methane with heat sources both internal and external to the engine was considered. In general, small gains in performance were obtained for the various cycles with respect to a reference turbojet cycle. No attempt was made to determine the weights of the various cycles considered.

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SUMMARY

The performance of several methane-fueled engine cycles suitable for Mach 3.0 flight is analyzed. The heat sink capacity of the fuel is used for turbine cooling as well as for other purposes. The primary figures of merit used for judging engine perform- ance are specific impulse, specific thrust, and thrust per unit compressor frontal area. In accord with the preliminary nature of this study, no attempt is made to determine en- gine weights associated with the various cycles.

The values of the performance parameters obtained for a methane-fueled turbojet are used as references in discussing other cycles. In the reference turbojet, turbine cooling is accomplished by passing fuel through the turbine blades.

In the first cycle considered, the methane fuel is preheated with a nonengine heat source before injection into the burner. This increased specific impulse 3.2 percent with respect to the reference cycle. A variation of the fuel preheat cycle, using a tur- bine to extract useful work and cool the fuel allowing further preheating, increased spe- cific impulse 4.5 percent. A cycle using the fuel heat sink to cool the airflow into the compressor yielded improvements in thrust per unit compressor frontal area of 8 per- cent at Mach 3.0 and 13.6 at takeoff. In the final cycle considered, the engine compres- sor is driven by heating and expanding the methane through a turbine. This offers a po- tential saving in engine weight with respect to the reference turbojet at the expense of specific impulse.

INTRODUCTION

Efficient flight at supersonic speeds depends on having engines with a high specific impulse and light weight. These characteristics depend on the engine cycle and fuel. Studies have indicated (ref. 1) that a methane-fueled turbojet offers about a 13 percent

improvement in specific impulse over a JP-fueled turbojet.

In addition, the large heat sink of methane, if used for turbine cooling, permits higher turbine inlet temperatures for a given turbine blade metal temperature. Consequently, a smaller engine is required for a specified thrust. Smaller engines usually weigh less than larger ones.

While the conventional turbojet cycle can take advantage of the higher heating value of methane and its heat sink capacity through turbine cooling, still other cycles can be devised to take advantage of these and other properties of the fuel. These additional methane properties of importance are its high specific heat, good thermal stability at high temperatures, and low boiling point. These characteristics make methane a relatively good working fluid. Of primary interest here is the application of the available fuel heat sink to other jobs in addition to turbine cooling.

Selected properties of methane (refs. 1 and 2) are as follows:

Boiling point (1 atm), ^{O}R (K) 201 (112)

Density (liquid), lb/cu ft (kg/cu m) 26 (416)

Freezing point (1 atm), ^{O}R (K) 163 (90. 6)

Lower heating value, Btu/lb (J/kg) . . . . 21 200 (4. 92×10^7)

Heat sink (initially liquid at 201° R (112 K)), Btu/lb (J/kg)

    at 1460° R (811 K) . . . . 1100 (2. 55×10^6)

    at 2000° R (1111 K) . . . . 1630 (3. 78×10^6)

Heat of vaporization (1 atm), Btu/lb (J/kg) . . . . 219 (5. 09×10^5)

Specific heat (liquid), Btu/lb ^{O}R (J/kg K) 0. 82 (3429)

Stoichiometric fuel-air ratio 0. 0581

Spontaneous ignition temperature, ^{O}R (K) 1630 (905. 6)

For this analysis, the quantities of most interest are the fuel's lower heating value (enthalpy of combustion with oxygen with resultant water considered as a vapor) and heat sink capacity.

The heating value of methane is about 13 percent greater than that of JP. This accounts for the increased specific impulse as compared to JP. Although the low initial temperature of methane contributes to its excellent heat sink capacity, it detracts from the fuel total enthalpy (enthalpy of combustion plus sensible enthalpy of fuel before combustion) that would be observed were the fuel vaporized and preheated to a higher temperature. However, this can occur when the fuel heat sink is used. Thus, in effect, the heat sink may be used to increase the fuel's total enthalpy yielding a consequent increase in specific impulse when the preheating is accomplished with a nonengine heat source (external preheating). Another possible use of the fuel heat sink is compressor precooling to increase engine airflow, thrust, and specific impulse.

The working fluid properties of methane can be used to advantage by expanding the fuel through a turboexpander to partially unload the main turbojet turbine. An expansion of the fuel in a turboexpander increases the heat sink available to the cycle and, hence, the possibility of further increasing the fuel total enthalpy.

In this analysis, a number of engine cycles are compared on the basis of specific impulse, specific thrust, and thrust per unit compressor frontal area. The rather preliminary nature of this study precludes the determination of actual engine and system weights. The cycle performance comparisons are made at Mach 3.0 in the stratosphere and, in one case, at sea-level static conditions. The performance of a methane-fueled turbojet cycle with direct fuel cooling of the turbine was used as a reference.

SYMBOLS

A area, sq ft; sq m

A_{tb} turbine area requiring cooling, sq ft; sq m

C_p gas specific heat at constant pressure, Btu/(lb)(^{O}R); J/(kg)(K)

F thrust, lb; kg

$F/\dot{W}_a$ specific thrust, sec (see eq. (5))

f fuel-air ratio

g acceleration of gravity, ft/sec^2; m/sec^2

h heat transfer coefficient, Btu/(ft^2)(sec)(^{O}R); J/(m^2)(sec)(K)

I fuel specific impulse, sec (see eq. (6))

L/D aircraft lift to drag ratio

N compressor rotational speed, rpm

P total pressure, lb/ft^2; N/cm^2

Q heat flow, Btu/sec; J/sec

R Breguet cruise range, ft; m

T total temperature, ^{O}R; K

V velocity, ft/sec; m/sec

W weight, lb; kg

$\dot{W}$ weight flow, lb/sec; kg/sec

WG   aircraft weight at start of cruise, lb; kg

γ ratio of specific heats

δ ratio of total pressure to standard sea-level pressure, $P_1/2116$; $P_1/10.13$

η adiabatic efficiency, percent

θ ratio of total temperature to standard sea-level temperature, $T_1/518.7$; $T_1/288$

Subscripts:

A   heat exchanger A

a   air

B   heat exchanger B

c   compressor

E   exhaust

e   engine

f   fuel

t   turbine

tb   turbine blade

0   free stream

1   compressor inlet

2   compressor exit

3   turbine inlet

4   turbine exit

Superscripts:

n   engine weight exponent

*   denotes change in quantity from reference value

METHOD OF ANALYSIS

Figures of Merit

The present analysis considers the performance of a number of methane-fueled en-

gine cycles that may be applicable to supersonic cruise aircraft. The primary figures of merit used herein for judging engine performance are specific impulse I, specific thrust $F/\dot{W}_a$, and thrust per unit compressor frontal area F/A_c. These were evaluated at Mach 3.0 in the stratosphere and, in one case, at sea-level static conditions.

The relation of the engine performance parameters chosen to aircraft performance can be shown by considering cruise range. The Breguet range equation

$$R = I V \frac{L}{D} \ln\left(\frac{WG}{WG - W_f}\right) \tag{1}$$

shows that range is proportional to specific impulse. Thus, for instance, in a simplified analysis, a 1 percent improvement in cruise specific impulse leads to a 1 percent improvement in cruise range. This implies that the product of cruise velocity and cruise lift to drag ratio remains constant and that the aircraft weight at the start of cruise and cruise fuel also are constant.

Although no attempt was made to calculate the weights associated with each engine cycle considered, the effect of changes in specific thrust and thrust per unit compressor frontal area can be shown by assuming a weight scaling law. To illustrate this, a scaling law is used that assumes engine weight varies with compressor frontal area as follows:

$$\frac{W_e^*}{W_e} = \left(\frac{A_c^*}{A_c}\right)^n \tag{2}$$

For a constant engine thrust, (i.e., $F = F^*$) the equation may be rewritten as:

$$\frac{W_e^*}{W_e} = \left[\frac{\left(\frac{F}{A_c}\right)}{\left(\frac{F}{A_c}\right)^*}\right]^n = \left[\frac{\left(\frac{F}{\dot{W}_a}\right)\left(\frac{\dot{W}_a}{A_c}\right)}{\left(\frac{F}{\dot{W}_a}\right)^*\left(\frac{\dot{W}_a}{A_c}\right)^*}\right]^n \tag{3}$$

Equation (3) can accommodate both changes in specific thrust and thrust per unit compressor frontal area. If aircraft payload as well as aircraft gross weight at the start of cruise and cruise thrust are assumed constant, the Breguet range equation may be written as follows to show the effect of changing engine weight.

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$$R = I V \frac{L}{D} \ln \left[\frac{WG}{WG - W_f - W_e \left(1 - \frac{W_e^*}{W_e}\right)} \right] \tag{4}$$

Thus, for example, this equation, along with equation (3), shows that if engine thrust per unit compressor frontal area is increased from F/A_c to $(F/A_c)^*$, an increase in cruise range is obtained. A similar statement applies to specific thrust. However, if the specific impulse is reduced in obtaining these increases, the range may or may not be improved.

The preceding discussion shows how cruise range is affected by the engine performance parameters investigated. In general, the most useful aircraft performance parameters will depend on the particular aircraft and mission considered. Range was used in this discussion as an example.

Basic Cycle Relations

The basic engine cycle performance parameters of interest here are the specific thrust $F/\dot{W}_a$

$$\frac{F}{\dot{W}_a} = \frac{(1 + f)V_E - V_0}{g} \tag{5}$$

(where the exhaust flow is fully expanded) and specific impulse I

$$I = \frac{F}{\dot{W}_f} = \frac{F}{\dot{W}_a} \frac{1}{f} \tag{6}$$

The quantities required in these relations are the exhaust velocity V_E and the fuel to air ratio f. These are obtained from an analysis of the engine cycle. The basic procedures and relations used in calculating cycle performance are valid for all the cycles studied herein and may be found in reference 3.

Compression and expansion processes were analyzed using average values of the gas properties γ and C_p. For instance, the compressor exit temperature T_2 given by

$$T_2 = \frac{T_1}{\eta}\left[\left(\frac{P_2}{P_1}\right)^{(\gamma-1)/\gamma} - 1\right] + T_1 \tag{7}$$

was found for the γ corresponding to the temperature $T = (T_1 + T_2)/2$ by an iteration process. Similarly, the power required to drive the compressor, given by

$$\text{Compressor power} = \dot{W}_a C_p (T_2 - T_1) \tag{8}$$

was found for an average C_p. Expansions through the turbine and the fully expanded nozzle were treated in a similar manner.

The method and tables of reference 4 were used to perform thermodynamic calculations involving the combustion of fuel with air. The quantities obtained here are the fuel air ratio f and the gas properties C_p and γ.

Turbojet Cycle

The assumed component characteristics of the turbojet cycle are presented in table I for Mach 3.0 flight in the stratosphere and for sea-level static conditions. The cycle compressor map is shown in figure 1. The sea-level static design compressor pressure ratio is 9.9. For operation at a constant mechanical speed N the same compressor yields a pressure ratio of 3.5 at Mach 3.0 in the stratosphere. The corresponding compressor efficiencies are, respectively, 87 and 83.2 percent.

Although the comparative nature of this study makes the selection of engine char-

TABLE I. - TURBOJET ENGINE CHARACTERISTICS

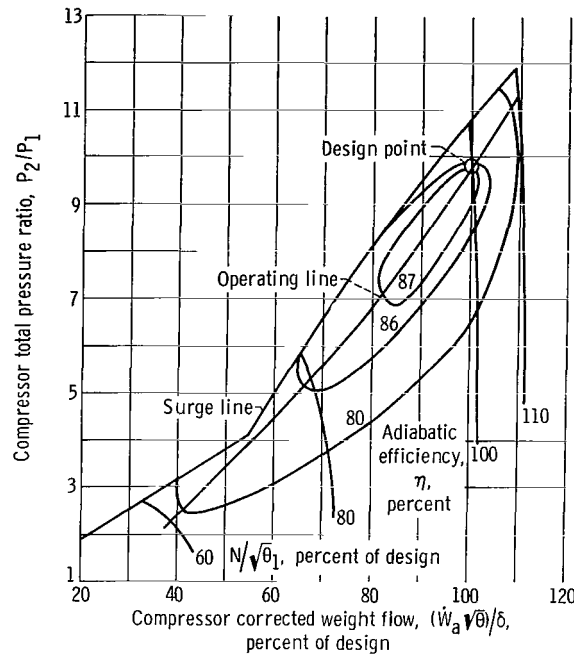| Characteristic	Mach 3.0 stratosphere	Sea-level static
Inlet pressure recovery	0.85	0.95
Compressor pressure ratio	3.5	9.9
Compressor adiabatic efficiency	0.832	0.87
Corrected weight flow, percent	51	100
Combustor pressure ratio	0.94	0.94
Combustor efficiency	0.98	0.98
Turbine efficiency	0.867	0.867
Nozzle thrust coefficient	0.98	0.98

Figure 1. - Turbojet compressor map.

acteristics somewhat arbitrary, the values selected are typical of those that yield good airplane performance considering both takeoff and climb as well as sustained Mach 3.0 cruise. Even though afterburning may be required during the transonic climb and acceleration phase of flight, the engine performance presented in the present analysis assumes maximum dry thrust at takeoff and during cruise.

Turbine cooling. - Turbine cooling was considered for the turbojet engine cycle. Turbine inlet gas temperature T_3 was varied over the range 2200° R (1222 K) to 3260° R (1811 K) with a heat input to the turbine given by the expression:

$$Q_{tb} = hA_{tb}(T_3 - T_{tb}) \tag{9}$$

The cooling load was found by assuming that a turbine blade metal temperature T_{tb} of 2200° R (1222 K) is acceptable.

Two methods of accomplishing turbine cooling are considered. The first method discussed is shown by the turbojet cycle of figure 2. Here turbine cooling is accomplished with compressor bleed air. The fuel heat sink is used to cool the bleed air in order to reduce the amount of compressor bleed air required. No bleed is required if the turbine inlet gas temperature is equal to or less than the allowable turbine metal temperature of 2200° R (1222 K). Detailed turbine cooling studies of the type of reference 5 indicate that a compressor bleed flow of approximately 6 percent is required at a
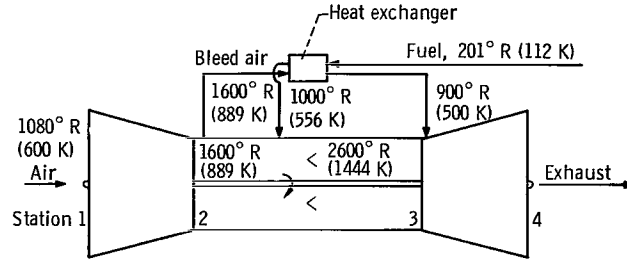
Figure 2. - Turbojet using compressor bleed air and fuel heat sink for turbine cooling. Typical Mach 3.0 cycle temperatures are shown.

turbine inlet temperature of 2660° R (1478 K). Taking this as a reference point, the schedule of compressor bleed flow with turbine inlet gas temperature shown in figure 3 was obtained by taking a linear variation of bleed air with temperature difference above the allowable blade metal temperature of 2200° R (1222 K). At a turbine inlet temperature of 2600° R (1444 K), figure 3 shows that a bleed flow of approximately 5 percent is required for turbine cooling.

The second method considered for accomplishing turbine cooling is shown by the turbojet cycle of figure 4. This cycle uses the fuel heat sink in a turbine blade heat exchanger to cool the turbine. No compressor bleed air is required. It was estimated that a turbine inlet temperature of 3260° R (1811 K) will require all the heat sink available in heating the methane from a liquid at 201° R (112 K) to a gas at 1460° R (811 K). The percent of the heat sink required at other turbine inlet temperatures was then found as follows.

At equilibrium operating conditions, the heat absorbed by the turbine blades Q_{tb} must equal the heat absorbed by the fuel Q_f. The heat absorbed by the fuel is proportional to the fuel flow rate and the percent of the available heat sink used for turbine cooling.

$$Q_{tb} = Q_f \propto \dot{W}_f \text{ (Percent heat sink)} \tag{10}$$

The engine fuel consumption rate is proportional to the airflow and the temperature increase in the combustor.

$$\dot{W}_f \propto \dot{W}_a(T_3 - T_2) \tag{11}$$

The turbine area cooled A_{tb} was assumed to be proportional to the turbine annulus area. For a constant flow Mach number, this area is proportional to airflow and the
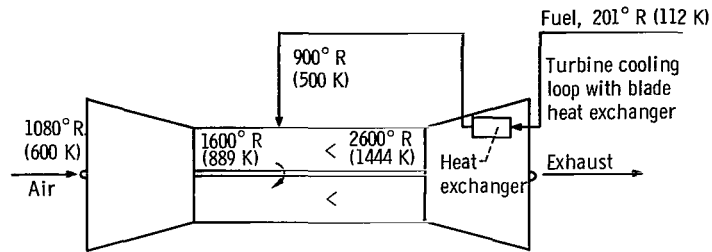
Figure 4. - Reference turbojet cycle using fuel heat sink for turbine cooling. Typical Mach 3.0 cycle temperatures are shown.
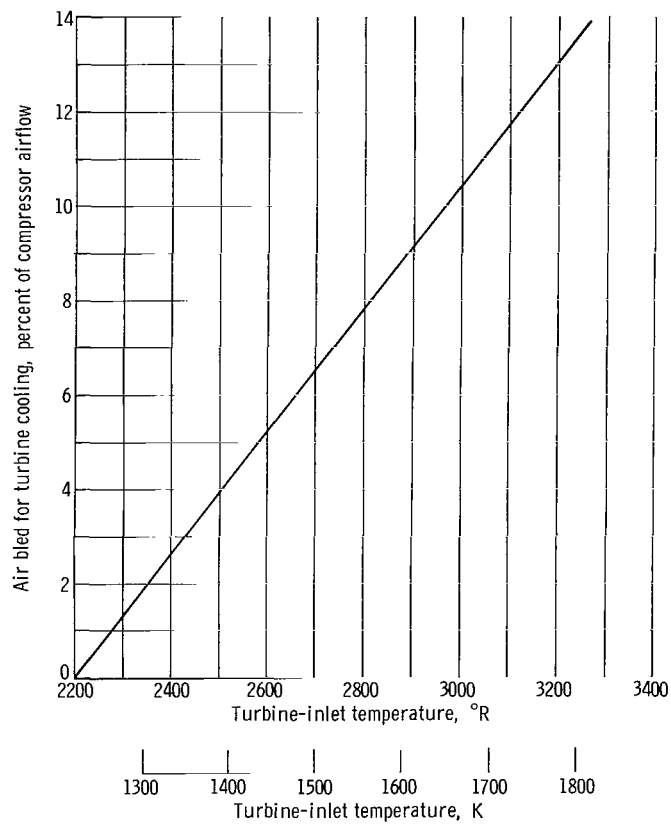

Figure 3. - Turbojet compressor bleed air required for turbine cooling. Blade metal temperature, 2200° R (1222 K) (see fig. 2).

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square root of temperature. Thus,

$$A_{tb} \propto \dot{W}_a (T_3)^{1/2} \tag{12}$$

Equation (10) can be solved with the aid of equations (9), (11), and (12) to give the fuel heat sink required for turbine cooling at any turbine inlet temperature.

(a) Maximum fuel preheat temperature, 1460° R (811 K).

(b) Maximum fuel preheat temperature, 2000° R (1111 K).
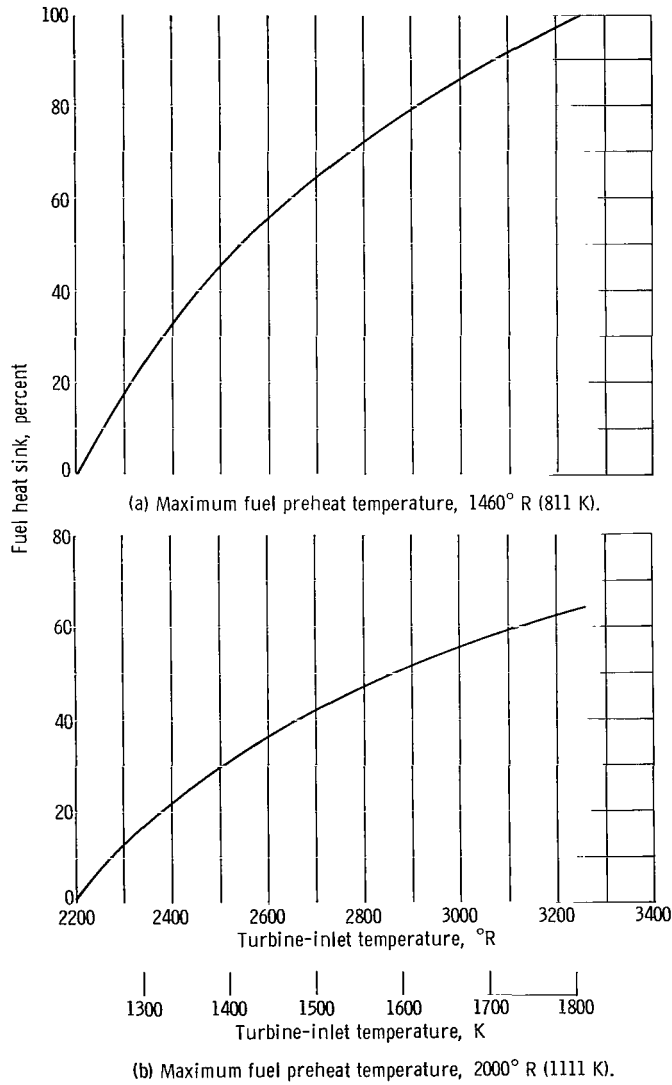
Figure 5. - Percentage of fuel heat sink required for reference turbojet turbine cooling. Mach 3.0 in stratosphere; blade metal temperature, 2200° R (1222 K); initial methane temperature 201° R (112 K) (see fig. 4).

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$$\text{Percent heat sink} \propto \frac{(T_3)^{1/2}(T_3 - T_{tb})}{(T_3 - T_2)} \tag{13}$$

The proportionality constant may be found since it was estimated that a turbine inlet temperature of 3260° R (1811 K) will require 100 percent of the heat sink available in heating the methane from a liquid at 201° R (112 K) to a gas at 1460° R (811 K).

Figure 5(a) shows the percent of the fuel heat sink required for turbine cooling versus turbine inlet temperature for a maximum methane preheat temperature of 1460° R (811 K). The total heat sink available is 1100 Btu per pound of fuel (2.55×10^6 J/kg) if the fuel is initially a liquid at 201° R (112 K). Figure 5(b) shows the percent of the fuel heat sink required to cool the turbine for a maximum methane preheat temperature of 2000° R (1111 K) where the heat sink available is 1630 Btu per pound of fuel (3.78×10^6 J/kg). The limiting temperatures of 1460° R (811 K) and 2000° R (1111 K) were arbitrarily chosen to indicate the sensitivity of cycle performance to changes in maximum permissible fuel preheat temperature. An acceptable fuel decomposition rate would be a major factor in selecting the practical operating limits.

Some of the special cycles to be discussed in the RESULTS AND DISCUSSION require modifications to the methods previously described. These modifications are noted in the description of the special cycles.

RESULTS AND DISCUSSION

Performance of Turbojet Cycle

Figure 6 presents the performance of several turbojet cycles at Mach 3.0 in the stratosphere. Engine performance, in terms of specific impulse and specific thrust, is shown for a range of turbine inlet temperatures from 2200° R (1222 K) to 3260° R (1811 K). The upper line with the open symbols gives the performance of the turbojet cycle with direct fuel cooling of the turbine, the cycle shown in figure 4. For this cycle, the peak specific impulse occurs at a turbine inlet temperature of approximately 2600° R (1444 K). The corresponding values of specific impulse and specific thrust are, respectively, 2790 and 40.1 seconds. These values are used as a reference for comparison with the other cycles investigated.

The closed symbols on the upper line show the turbojet performance if no turbine cooling is required. Compared to this case, turbine cooling directly with the fuel (the upper curve with open symbols) does not change the variation of specific impulse with specific thrust. However, turbine cooling does cause the peak specific impulse to occur
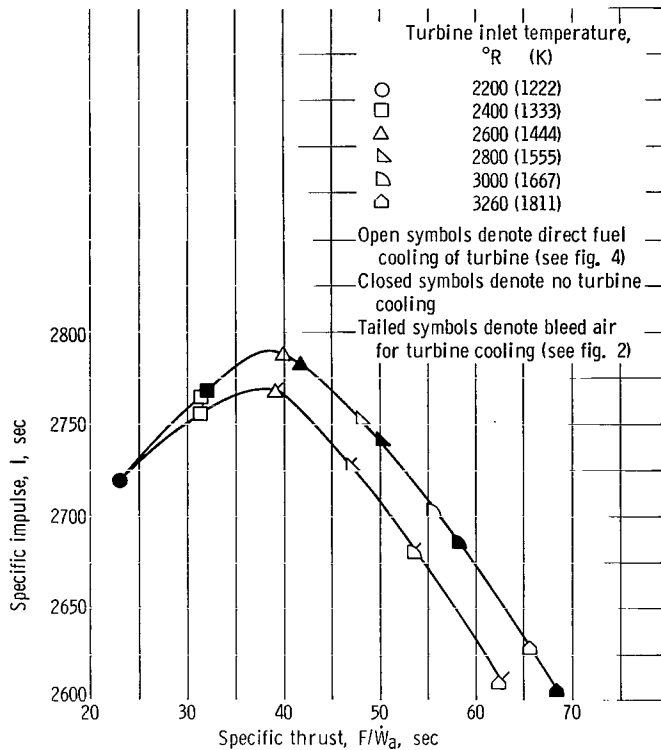
Figure 6. - Turbojet cycle performance with several methods of turbine cooling. Mach 3.0 in stratosphere.

at a slightly higher turbine inlet temperature. If turbine inlet temperature is fixed, turbine cooling causes a slight reduction in specific thrust. In practice, the elimination of turbine cooling would require a substantial improvement in turbine materials.

The lower curve of figure 6 with tailed symbols shows the performance of the turbojet cycle where the fuel heat sink is used to cool compressor bleed air which, in turn, cools the turbine, the cycle of figure 2. With this cycle, the specific impulse is about 1 percent less and the specific thrust is about 1.2 percent less than the reference cycle at 2600° R (1444 K). This cycle, using compressor bleed air, was used for the airplane performance calculations of reference 1.

Turbojet Cycle With External Preheating of the Fuel

The turbojet cycle with external preheating of the fuel is shown in figure 7. It differs from the reference turbojet cycle of figure 4 by the addition of heat exchanger A to the fuel system. This section and following sections discuss engine cycles which take advantage of methane's heat sink, thermal stability, and ability to be vaporized and used
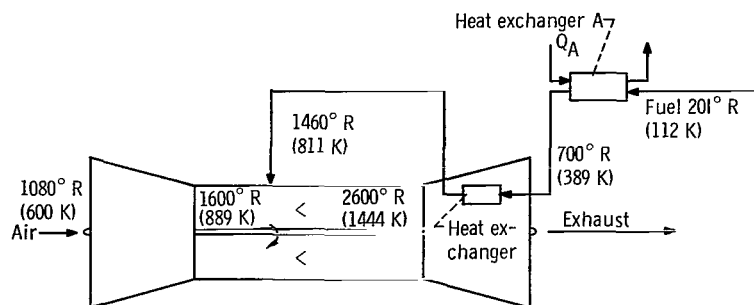
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Figure 7. - Turbojet cycle with external preheating of fuel. Typical Mach 3.0 cycle temperatures are shown.

as a working fluid for other purposes in addition to turbine cooling.

Heat sources. - The cycle discussed in this section and cycles to be discussed later consider preheating of the fuel with a heat source external to the engine. Several sources of heat external to the engine are available. For example, at the Mach 3.0 cruise condition, stagnation temperatures on the order of 1080° R (600 K) are encountered. Thus, aerodynamic heating of the fuel could possibly be obtained by placing heat exchangers at strategic locations in the airframe. Other potential heat sources commonly requiring cooling could also be used. Several of the major sources include hydraulic fluid and pumps, electrical equipment, and the cabin or crew area. In light of the sources available, it was assumed that, if desired, sufficient energy could be obtained from sources external to the engine to preheat the fuel from a liquid at 201° R (112 K) to a gas at 1000° R (556 K). If this temperature rise is obtained from aerodynamic heating, the required heat exchanger effectiveness (defined as obtained temperature rise divided by maximum theoretical temperature rise) is 91 percent.

Temperatures in excess of 1000° R (556 K) can be obtained by cooling the hot parts of the engine. Energy obtained in this manner was charged to the cycle. Unless stated otherwise, the component characteristics of the cycles to follow are identical to those of the reference turbojet cycle (see table I and fig. 1).

Effect of turbine cooling on fuel preheating. - The heat that may be added to the fuel is not only a function of the external source temperature but (when turbine cooling is considered) also of the turbine inlet temperature and the limiting maximum fuel preheat temperature.

The relation of turbine cooling to the degree of external fuel preheating possible can be seen by referring again to figure 5(a), which is for a maximum methane preheat temperature of 1460° R (811 K). For example, at a turbine inlet temperature of 2600° R (1444 K), 56 percent of the available heat sink is required for turbine cooling. The turbine cooling can be done with the higher temperature end of the fuel's heat sink capacity. The additional heat Q_A that the fuel may absorb is the remaining 44 percent of the heat

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sink or, in this case, 484 Btu per pound of fuel (1. 12×10^6 J/kg). This enthalpy change can be obtained by externally preheating the fuel from a liquid at 201° R (112 K) to a gas at 700° R (389 K). Hence, even though an external source is available at 1000° R (556 K) when the aircraft is cruising at Mach 3.0, total heat sink considerations and turbine cooling requirements restrict the allowable fuel external preheat temperature to 700° R (389 K) for this example.

The procedure outlined in the previous paragraph was used to obtain the allowable methane external preheat temperatures shown in figure 8. This figure shows that for a maximum methane preheat temperature of 1460° R (811 K) the fuel may be externally preheated to 1000° R (556 K) only if the turbine inlet temperature is equal to or less than 2430° R (1350 K). As turbine inlet temperature is increased, turbine cooling requirements reduce the permissible external preheat temperature from 1000° R (556 K) at the turbine inlet temperature of 2430° R (1350 K) to 201° R (112 K) at a turbine inlet temperature of 2910° R (1617 K). As the turbine inlet temperature is further increased toward 3260° R (1811 K), the permissible heat input to the fuel from heat exchanger A is reduced to the point where less than 100 percent of the fuel is vaporized; thus, there is no fuel temperature rise in this region.

If the maximum methane preheat temperature is increased from 1460° R (811 K) to 2000° R (1111 K), figure 8 shows that the fuel may now be externally preheated to 1000° R
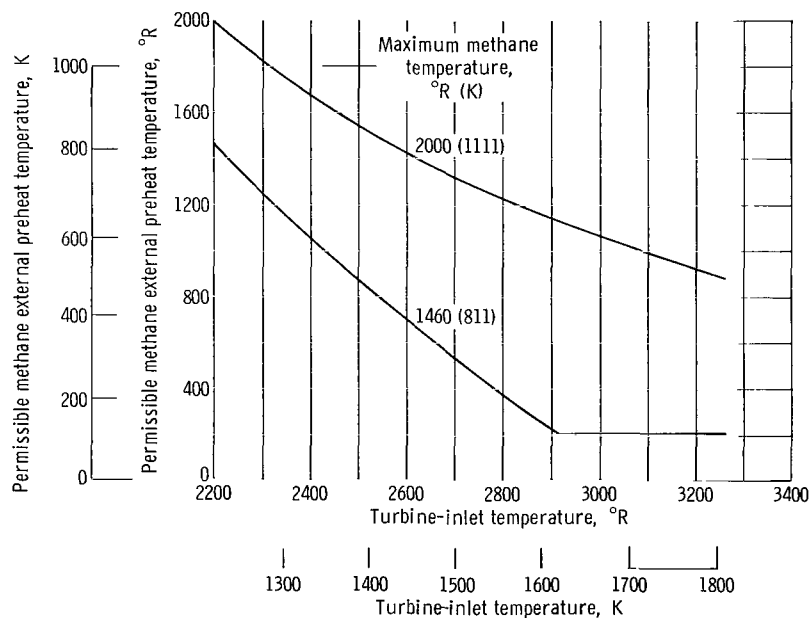

Figure 8. - Effect of turbine-inlet temperature on permissible methane external preheat temperature. Turbine cooling to maintain 2200° R (1222 K) blade metal temperature; Mach 3.0 in stratosphere.

(556 K) at turbine inlet temperatures as high as 3100° R (1722 K). For both maximum fuel preheat temperatures, the external energy that may be added to the fuel becomes identical to its heat sink capacity at a turbine inlet temperature of 2200° R (1222 K) where no turbine cooling is required. If no upper limit is set on the permissible fuel preheat temperature, it may be preheated to 1000° R (556 K) for all turbine inlet temperatures considered.

Effect of fuel preheating. - The increased total enthalpy of the fuel obtained by this external preheating will reduce the fuel-air ratio f and, consequently, as shown by equation (6), increase the engine specific impulse I. The percent increase in specific impulse over that of the reference turbojet obtained by external preheating of the fuel is shown in figure 9. The turbine inlet temperature was fixed at 2600° R (1444 K). The vertical segment of the curve (constant preheat temperature of 201° R (112 K)) shows the
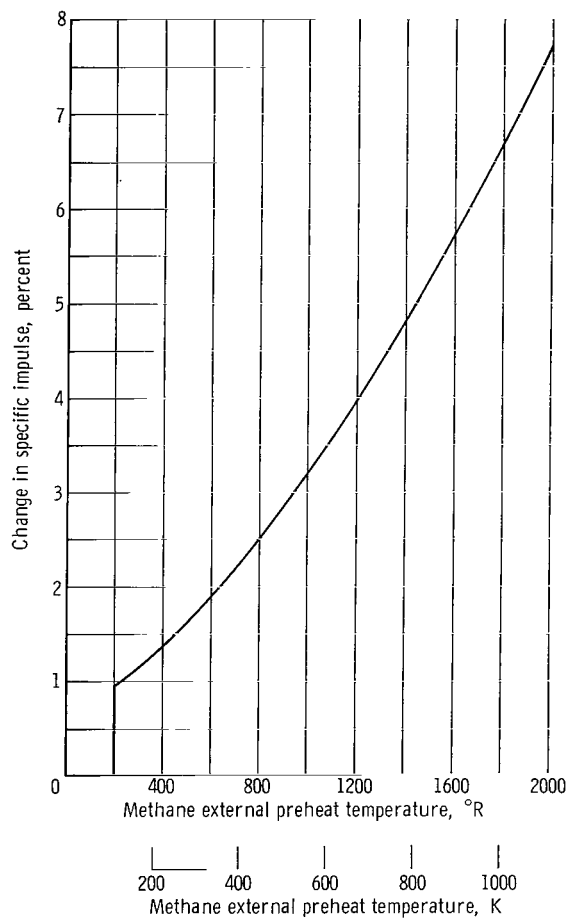

Figure 9. - Increase in specific impulse obtained by preheating methane fuel with heat source external to engine. Reference specific impulse, 2790 sec; turbine-inlet temperature, 2600° R (1444 K); Mach 3.0 in stratosphere.

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specific impulse gain obtained by adding enough external heat to vaporize the fuel. External preheating of the fuel to 700° R (389 K), the temperature consistent with the turbine cooling requirement and a 1460° R (811 K) maximum methane temperature, yields a gain in specific impulse of 2.2 percent. If the maximum methane preheat temperature is increased from 1460° R (811 K) to 2000° R (1111 K), the allowable external preheat temperature to satisfy the turbine cooling requirement becomes 1420° R (789 K). This is more than adequate to allow the fuel to be preheated to the 1000° R (556 K) limit assumed to be imposed by the available external source. In this case, a 3.2 percent gain in specific impulse is possible (fig. 9). Not shown is a loss in specific thrust and thrust per unit compressor frontal area of approximately 0.2 percent due to the reduced fuel flow and, hence, engine mass flow obtained with the lower fuel-air ratio.

If the constraints of a limiting maximum fuel preheat temperature and limited external source temperatures are removed, figure 9 shows that external preheating of the fuel to 2000° R (1111 K) could yield a specific impulse gain of nearly 8 percent. Associated with this is a loss in specific thrust of approximately 1 percent. However, for the present Mach 3.0 study, an external source at this temperature is not available.

Effect of turbine inlet temperature. - The results of figure 9 were for a turbojet with a turbine inlet temperature of 2600° R (1444 K). Figure 10 presents engine performance, specific impulse versus specific thrust, of the turbojet with fuel preheating for a range of turbine inlet temperatures.

The lower line gives the performance of the reference turbojet cycle which uses di-
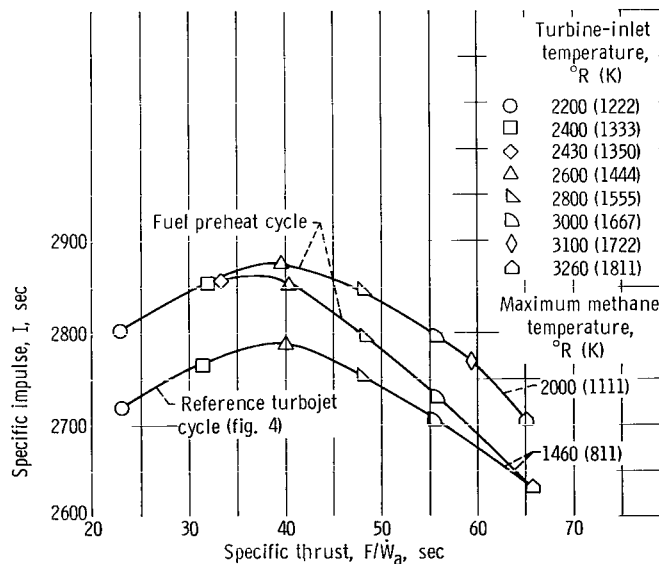

Figure 10. - Turbojet cycle performance obtained by externally preheating fuel with 1000° R (556 K) heat source. Mach 3.0 in stratosphere.

rect fuel cooling of the turbine and was previously discussed in regard to figure 6. The uppermost line shows the performance obtained by preheating the fuel with an external source with the maximum methane temperature limited to 2000° R (1111 K). Along this line, the fuel can be externally preheated to 1000° R (556 K) until a turbine inlet temperature of 3100° R (1722 K) is reached. Above this turbine inlet temperature, reduced preheating is required to accommodate turbine cooling. At the reference turbine inlet temperature of 2600° R (1444 K), a 3.2 percent gain in specific impulse is available.

The middle curve connecting the upper and lower curves of figure 10 shows the performance obtained with the preheat cycle with the maximum methane temperature limited to 1460° R (811 K). Below a turbine inlet temperature of 2430° R (1350 K), the fuel can be externally preheated to 1000° R (556 K) and turbine cooling then accomplished without exceeding the maximum fuel temperature of 1460° R (811 K). In this region, the performance thus coincides with that obtained for the case with a maximum fuel temperature of 2000° R (1111 K). Above a turbine inlet temperature of 2430° R (1350 K), reduced external preheating is required to accomplish turbine cooling. As the turbine inlet temperature is increased above 2430° R (1350 K), the performance of this cycle approaches that of the reference turbojet. At a turbine inlet temperature of 3260° R (1811 K) where all of the heat sink is required for turbine cooling, the performance of the turbojet cycle with external preheating of the fuel coincides with that of the reference turbojet. At the reference turbine inlet temperature of 2600° R (1444 K), a 2.2 percent gain in specific impulse is obtained with the maximum methane temperature limited to 1460° R (811 K).

If the maximum methane preheat temperature is increased to 1680° R (934 K), operation is possible along the uppermost curve of figure 10 to a turbine inlet temperature of 2600° R (1444 K). This yields near maximum specific impulse with a gain over the reference turbojet of 3.2 percent as quoted previously.

Turbojet Cycle With External Preheating of the Fuel Augmented by Fuel Expansion

The purpose of this cycle is to improve engine performance by adding more heat to the methane from the external sources while maintaining the same source temperature of 1000° R (556 K). The cycle is shown in figure 11 and differs from the one shown in figure 7 by the addition of a turboexpander and heat exchanger B to the fuel system. Although heat exchanger B is not required over the entire range of turbine inlet temperatures considered, it is included in the general discussion of the cycle and is required at low turbine inlet temperatures where turbine cooling requirements are small. At the reference turbine inlet temperature of 2600° R (1444 K), the cycle operates in the following manner.

Heat exchanger B

Heat exchanger A

Q_B

Q_A

1000° R (556 K)

1000° R (556 K)

Fuel, 201° R (112 K)

1460° R (811 K)

700° R (389 K)

1080° R (600 K)

1600° R (889 K)

2600° R (1444 K)
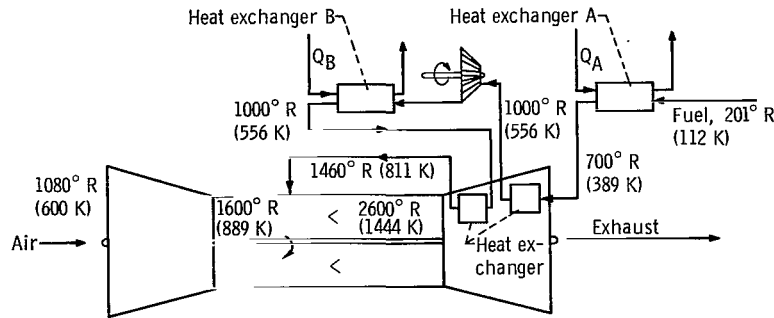
Heat exchanger

Exhaust

Air

Figure 11. - Turbojet cycle with external fuel preheating augmented by turboexpander. Typical Mach 3.0 cycle temperatures are shown.

The fuel is first pressurized as a liquid at 201° R (112 K) and sent to heat exchanger A where it is heated to 700° R (389 K) by the external source. It then passes through part of the turbojet turbine where it does some turbine cooling and is heated to 1000° R (556 K). Next it passes through a turboexpander which extracts useful work from the methane thereby cooling it below the expander inlet temperature of 1000° R (556 K). The methane then goes to heat exchanger B where it is again heated to 1000° R (556 K) by the external heat source. The last heat exchange is to complete the turbojet turbine cooling requirement. Here the methane is heated to the limiting temperature of 1460° R (811 K) used in this example.

Note that the temperature exiting the first turbojet turbine heat exchanger is the same as that entering the second turbojet turbine heat exchanger. Thus, the addition of the turboexpander and heat exchanger B has not changed the heat sink available for turbine cooling. However, the addition of the turboexpander permits the addition of the heat input Q_B to the cycle; and it is this heat that can improve cycle performance.

The useful power output of the turboexpander is, in the present analysis, used to partially drive the primary turbojet compressor and thus unload the turbine. In analyzing this cycle, the efficiency of the turboexpander was fixed at 85 percent.

Effect of turboexpander pressure ratio. - The percent improvement in engine specific impulse and specific thrust obtained with the augmented preheat cycle over that of the reference turbojet is shown in figure 12(a). The turbine inlet temperature is 2600° R (1444 K). Turboexpander pressure ratio was varied over the range 1 to 30. Performance is shown for both limiting methane temperatures of 1460° R (811 K) and 2000° R (1111 K). At a turboexpander pressure ratio of 1, this cycle reduces to the preheat cycle and the performance is the same as that shown in figure 10. As the turboexpander pressure ratio is increased, both the specific impulse and specific thrust increase. When the methane preheat temperature is limited to 1460° R (811 K), the lower specific impulse curve of figure 12(a), increasing the turboexpander pressure ratio from 1 to 30 increases
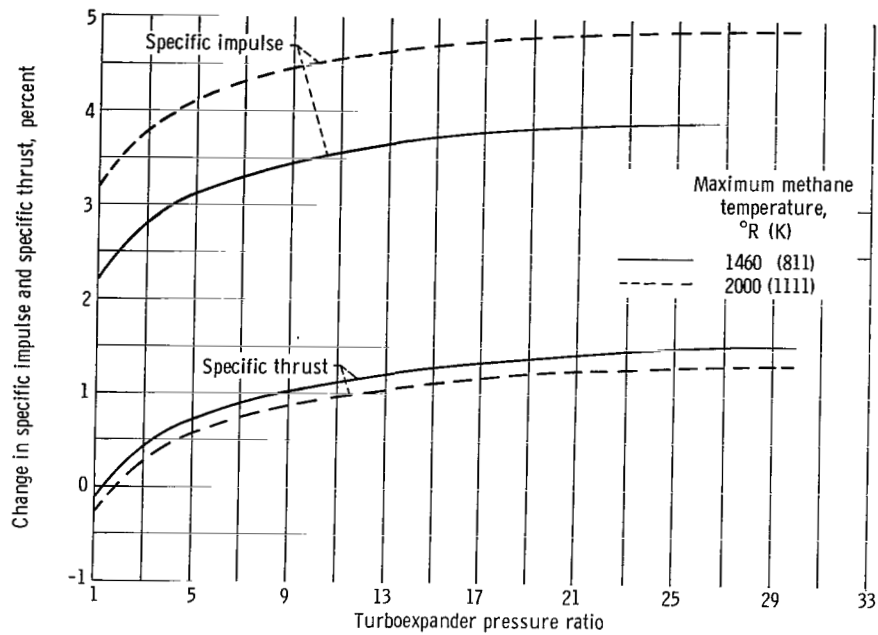
(a) Effect of turboexpander pressure ratio. Primary turbine-inlet temperature, 2600° R (1444 K); reference specific impulse, 2790 sec; reference specific thrust, 40.1 sec.
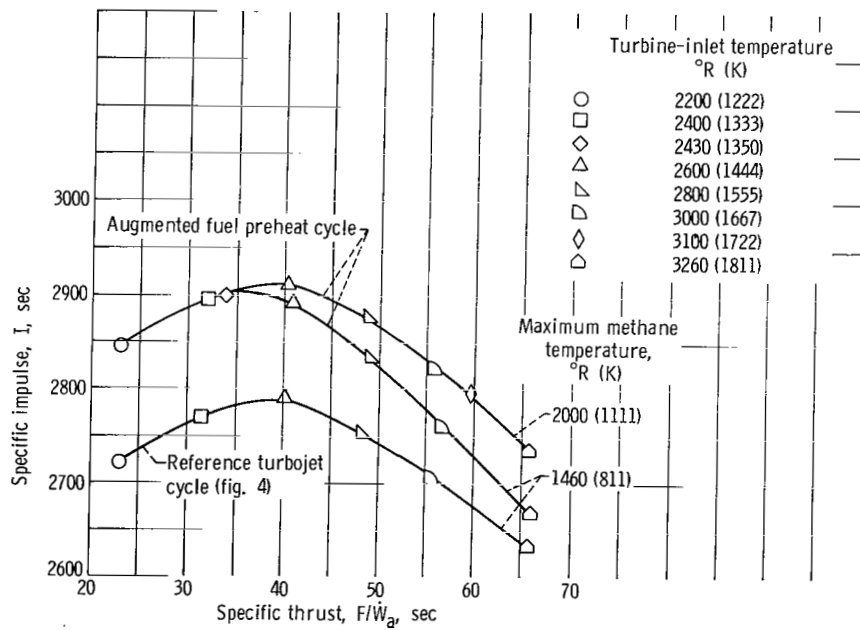
(b) Effect of primary turbine-inlet temperature. Turboexpander pressure ratio, 10.

Figure 12. - Performance of turbojet cycle with external fuel preheating augmented by turbo-expander. External heat source temperature and turboexpander inlet temperature, 1000° R (556 K); Mach 3.0 in stratosphere.

the specific impulse gain from 2.2 to 3.9 percent. The corresponding values of specific impulse gain for a limiting methane temperature of 2000° R (1111 K) are 3.2 and 4.9 percent. The increase in specific thrust is about 1.7 percent in each case.

The most rapid improvements in cycle performance with increases in turboexpander pressure ratio occur at the lower pressure ratios. A turboexpander pressure ratio of 10 was selected to show the effect of turbojet turbine inlet temperature on cycle performance. At this pressure ratio and Mach 3.0 cruise at 65 000 feet (19 815 m), the required fuel pressure into the turboexpander is 1200 pounds per square inch (827 N/cm^2).

Effect of turbine inlet temperature. - The performance of the augmented preheat cycle with a turboexpander pressure ratio of 10 is shown in figure 12(b) over a range of turbine inlet temperatures. Again the performance of the reference turbojet cycle, the lowest line, is given for comparison. The uppermost line shows the performance obtained with the augmented preheat cycle for a maximum methane temperature of 2000° R (1111 K). At the reference turbine inlet temperature of 2600° R (1444 K), a gain in specific impulse of 4.5 percent is obtained with respect to the reference turbojet. Specific thrust is increased 1.0 percent. In a manner analogous to that discussed in conjunction with figure 10, the energy that may be added to the fuel is reduced as turbojet turbine inlet temperature is increased. This can be illustrated by considering the performance of this cycle with the maximum fuel temperature limited to 1460° R (811 K). The resultant performance, as shown by the middle curve of figure 12(b), is reduced below that of the 2000° R (1111 K) methane temperature case as the turbine inlet temperature is increased above 2430° R (1350 K). At the reference turbine inlet temperature of 2600° R (1444 K), the gain in specific impulse is 3.5 percent as compared to the 4.5 percent gain obtained with the higher allowable methane temperature. Gains in performance are obtained with respect to the reference turbojet cycle over the entire range of turbine inlet temperatures due to the work extraction and cooling process in the turboexpander. This cooling permits additional heat to be added to the fuel at all turbine inlet temperatures considered.

The specific impulse gain of 4.5 percent can be obtained with this cycle at a turbine inlet temperature of 2600° R (1444 K) if a maximum methane temperature of 1680° R (934 K) is permitted.

Turbojet Cycle With Compressor Precooling

The compressor precooling cycles shown in figures 13(a) and (b) use a heat exchanger or series of heat exchangers ahead of the compressor to cool the inlet air prior to its entering the engine. Unlike the previous cycles considered, this cycle was analyzed both at Mach 3.0 cruise (fig. 13(a)) and sea-level static conditions (fig. 13(b)). At Mach 3.0, it is assumed that the methane coolant flow rate is equal to the fuel flow rate to the en-
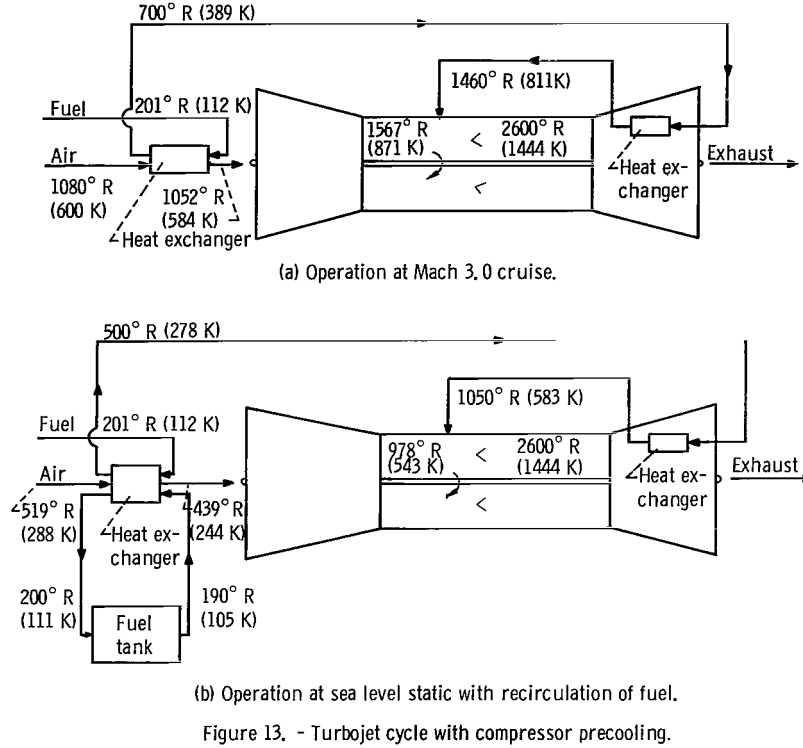
(a) Operation at Mach 3.0 cruise.

(b) Operation at sea level static with recirculation of fuel.

Figure 13. - Turbojet cycle with compressor precooling.

gine. At sea-level static conditions, increased fuel flow rates and, hence, additional cooling were obtained by recirculating fuel from and to the fuel tanks in addition to using the fuel as it flows to the engine.

For a constant compressor rotational speed N, a reduction in the compressor inlet temperature T_1 will lead to an increase in the corrected speed $N/(\theta)^{1/2}$. From figure 1, it is seen that, for the operating line used, this will increase both the compressor pressure ratio P_2/P_1 and the corrected weight flow $\dot{W}_a\,(\theta)^{1/2}/\delta$. The ratio of compressor weight flows for the precooled and the reference turbojet cycle is given by the expression:

$$\frac{\dot{W}_a^*}{\dot{W}_a} = \frac{\left(\dot{W}_a\dfrac{(\theta)^{1/2}}{\delta}\right)^*}{\dot{W}_a\dfrac{(\theta)^{1/2}}{\delta}}\left(\frac{T_1}{T_1^*}\right)^{1/2}\frac{P_1^*}{P_1} \tag{14}$$

where the pressure ratio through the heat exchanger is represented by the term P_1^*/P_1. The asterisk denotes the precooled cycle. This change in weight flow as a result of cool-

ing can be used to compute the percent change in thrust per unit compressor frontal area
by

$$\text{Percent change in } F/A_c = 100\left[\frac{\left(\dfrac{F}{\dot{W}_a}\right)^*}{\left(\dfrac{F}{\dot{W}_a}\right)}\left(\dfrac{\dot{W}_a^*}{\dot{W}_a}\right) - 1\right] \quad (15)$$

This quantity, in addition to specific impulse and specific thrust, is a figure of merit for
this cycle.

Operation at Mach 3.0. - At the Mach 3.0 cruise condition, only the fuel consumed
by the engine was passed through the heat exchanger. At a turbine inlet temperature of
2600° R (1444 K) with the methane limited to a maximum temperature of 1460° R (811 K),
sufficient heat sink is available to cool the compressor inlet air 28° R (15.5 K) from
1080° R (600 K) to 1052° R (584 K) (see fig. 13(a)). This increases the compressor
pressure ratio from 3.50 to 3.57. A small increase in compressor efficiency is also
obtained. In accomplishing this cooling, the fuel is vaporized and heated to 700° R
(389 K). This temperature permits the required turbine cooling to be obtained by further
preheating of the fuel to 1460° R (811 K).

The percent increase in cruise specific impulse, specific thrust, and thrust per unit
compressor frontal area obtained as a result of this 28° R (15.5 K) compressor inlet tem-
perature reduction is shown in figure 14(a). The change in performance is plotted against
the air side pressure loss through the air-fuel heat exchanger. For this degree of pre-
cooling, the method of reference 6 yields a theoretical pressure loss associated with heat
transfer of 0.4 percent. The theory uses Reynolds analogy to obtain a correlation be-
tween the heat transfer coefficient and the wall friction coefficient. At this pressure loss,
the improvements in specific impulse and specific thrust are 0.55 and 1.2 percent, re-
spectively. The most significant gain is an increase in thrust per unit compressor frontal
area of 4.8 percent. A heat exchanger total pressure loss of 4.3 percent would negate
this gain. Increases in specific impulse, specific thrust, and thrust per unit compressor
frontal area of 0.8, 1.7, and 8.0 percent, respectively, are obtained if the maximum al-
lowable methane temperature is increased to 1680° R (934 K). In this case, the methane
is preheated to 1000° R (556 K) in cooling the compressor airflow 40° R (22.2 K).

Sea-level static operation. - In addition to the Mach 3.0 flight condition, the perform-
ance of this cycle was also evaluated at sea-level static conditions. Two types of opera-
tion are considered. In the first instance, engine performance is obtained using only the
heat sink available in the fuel consumed by the engine. The cycle is thus the same as that
shown by figure 13(a). The second case obtains additional cooling by the addition of
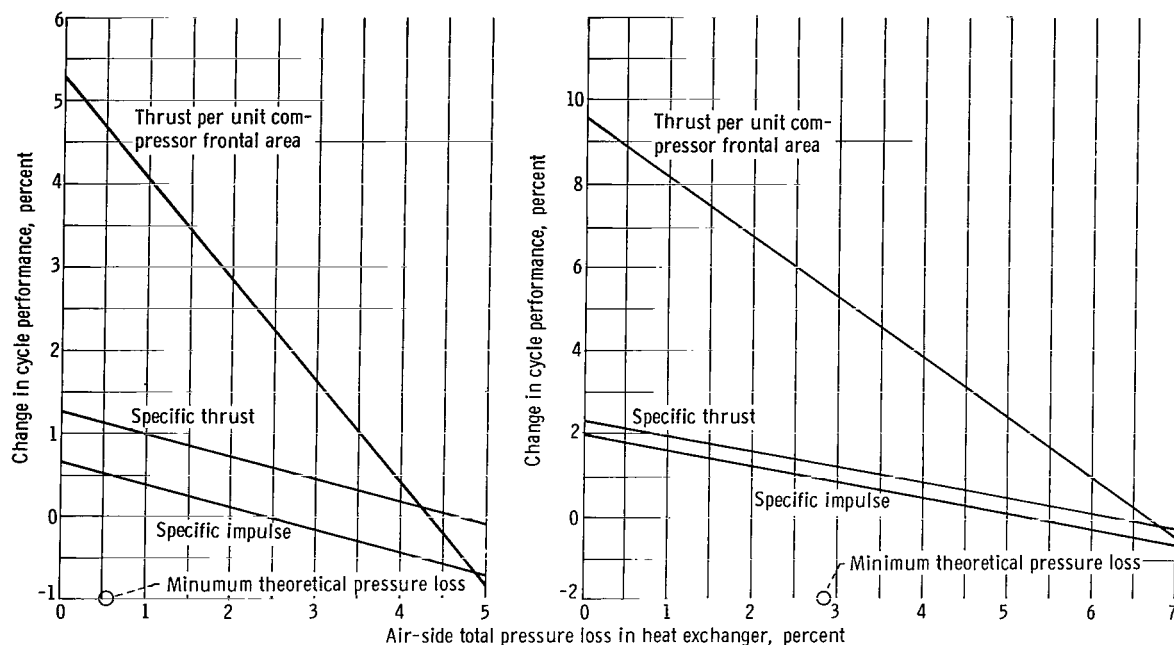
(a) Mach 3.0 in stratosphere. Compressor inlet air cooled, 28° R (15.5 K); no recirculation of fuel; reference specific impulse, 2790 seconds; reference specific thrust, 40.1 seconds.

(b) Sea-level static conditions. Compressor inlet air cooled, 33° R (18.4 K); no recirculation of fuel; reference specific impulse, 3990 seconds; reference specific thrust, 88.9 seconds.
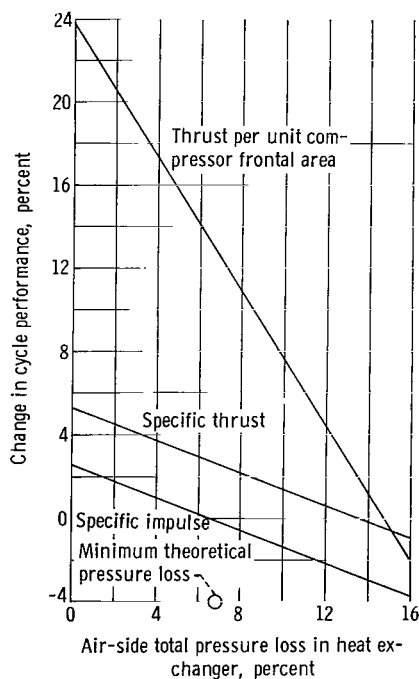
(c) Sea-level static conditions. Compressor inlet air cooled 80° R (44.5 K); recirculation of fuel; reference specific impulse 3990 sec; reference specific thrust, 88.9 sec.

Figure 14. - Effect of heat exchanger pressure loss on performance of turbojet cycle with compressor precooling. Turbine-inlet temperature, 2600° R (1444 K).

another fuel loop to the precooling heat exchanger. This case is shown by figure 13(b).
The increased cooling is obtained by recirculating a portion of the fuel from the tanks to
the heat exchanger and then back to the tanks again.

The sea-level static performance obtained by compressor precooling with the fuel
flow to the engine (no recirculation) is shown in figure 14(b). With an arbitrarily assumed
heat exchanger effectiveness of 94 percent, the fuel heat sink available can reduce the
compressor inlet temperature by 33° R (18. 4 K) at the reference turbine inlet tempera-
ture of 2600° R (1444 K). With a heat exchanger effectiveness of 80 percent, approxi-
mately 30° R (17 K) of precooling is possible. A reduction in inlet air temperature of
33° R (18. 4 K) increases the compressor pressure ratio from 9. 9 to 10. 4 and in doing so
causes a slight reduction in compressor efficiency. At the heat exchanger pressure loss
associated with heat transfer of 2. 9 percent (again obtained using Reynolds analogy,
ref. 6), an improvement in thrust per unit compressor frontal area of 5. 4 percent is
realized. Increases in specific impulse and specific thrust of 0. 9 and 1. 2 percent, re-
spectively, were also obtained.

Increasing the cooling by recirculation of the fuel leads to the performance shown in
figure 14(c). Here the precooling has been increased to 80° R (44. 5 K) yielding a com-
pressor pressure ratio of 11. 4. This precooling can be maintained with the aircraft of
reference 1 for approximately 1 minute if a 10° R (5. 5 K) increase in temperature of all
the liquid fuel is allowed. This, of course, requires an initial subcooling of the methane
below the normal boiling point of 201° R (112 K).

Again using the method of reference 6, a minimum heat exchanger pressure loss of
6. 6 percent was found for the 80° R (44 K) precooling. At this pressure loss, as shown
by figure 14(c), the 80° R (44. 5 K) temperature change produces a 13. 6 percent improve-
ment in thrust per unit compressor frontal area with respect to the reference turbojet.
The specific thrust is also increased 2. 8 percent while the specific impulse is equal to
that of the reference cycle. With this degree of precooling, a heat exchanger pressure
loss of nearly 15 percent would be required to completely negate the gain in thrust per
unit compressor frontal area.

Potential problem areas with this cycle are the possibility of ice formation on the
cold heat exchanger surfaces and the requirement for additional subcooling of the fuel.

Fuel Expansion Cycle

The cycles discussed to this point represent rather modest modifications to the ref-
erence turbojet cycle. Unlike these cycles, the fuel expansion cycles investigated in this
section constitute a basic change in engine design. Three versions of the fuel expansion
cycle are shown in figures 15(a), (b), and (c). In each case, the compressor is driven by

(a) Cycle variation A.

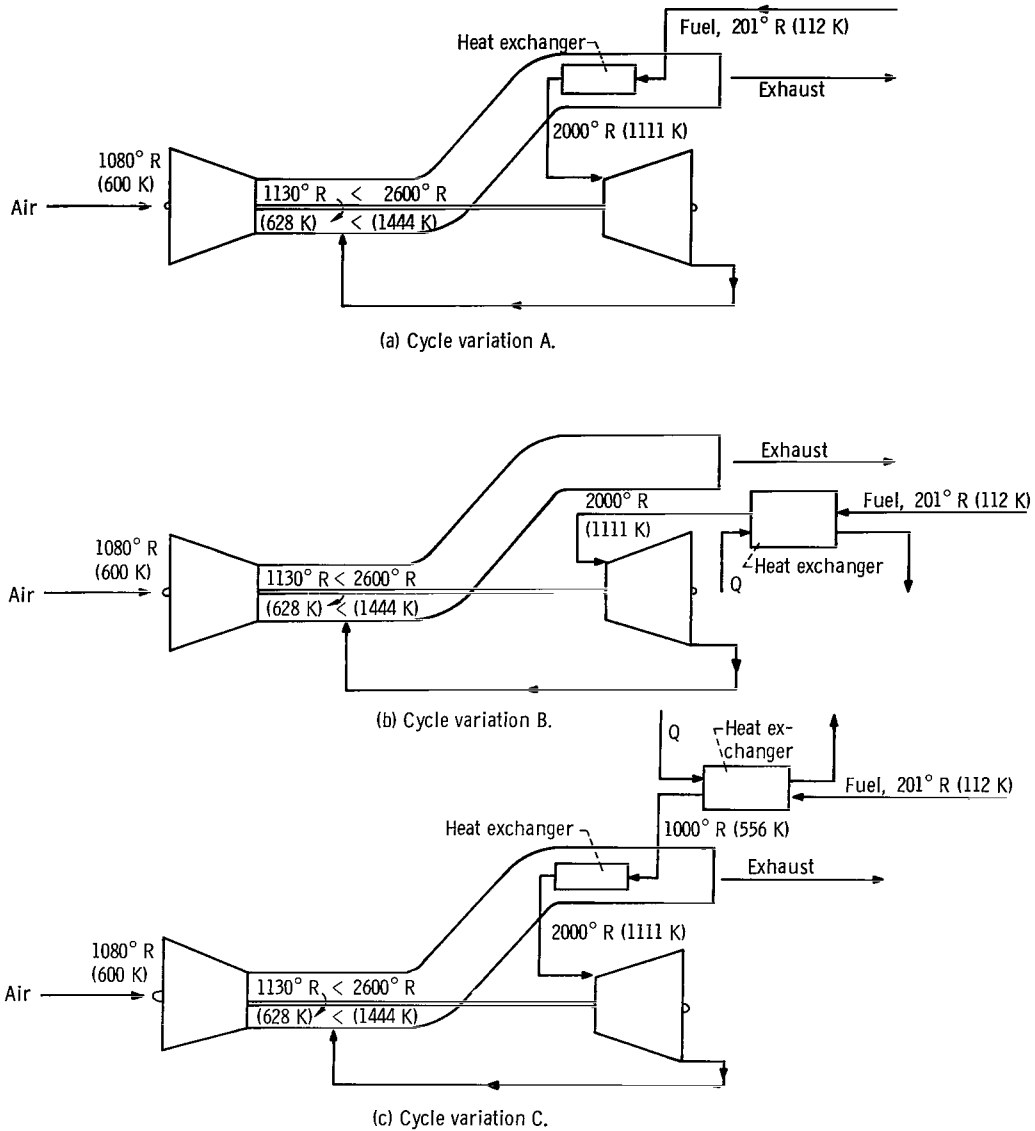(b) Cycle variation B.

(c) Cycle variation C.

Figure 15. - Fuel expansion cycle. Typical Mach 3.0 cycle temperatures are shown.

heating and expanding the methane fuel through a turbine. Therefore, these cycles differ from the augmented preheat cycle in that all the power required to drive the compressor, rather than just a portion of it, is obtained by expanding the heated fuel through a turbine. In addition, the fuel is not reheated after leaving the turbine. Here the compressor specific work, ΔH_c Btu/lb air (J/kg air) is related to the turbine specific work, ΔH_t Btu/lb fuel (J/kg fuel) by

$$\Delta H_c = \Delta H_t(f) \qquad (16)$$

Compressor and turbine efficiencies of 83 and 85 percent, respectively, were assumed for all operating conditions and performance was obtained for the Mach 3.0 flight condition.

The three fuel expansion cycles considered, referred to as variations A, B, and C (fig. 15), differ only in the method used to heat the fuel. Cycle variation A employs a heat exchanger to extract heat from the hot combustion gases. Thus, this cycle is wholly dependent upon the combustion of fuel as a heat source. Cycle variation B obtains all the heat required to heat the fuel from an external heat source. In cycle variation C, the fuel is heated by a combination of the two previous methods.

It must be pointed out that fuel expansion cycles, like compressor precooling cycles, have been studied in the past. For example, reference 7 describes a cycle (the air turboexchanger) that combines compressor precooling with the fuel expansion cycle variation A. In this reference, hydrogen was used as the fuel and working fluid. As an alternative to the heat exchanger of cycle variation A, the fuel can be heated by fuel-rich combustion in an auxiliary burner. This is essentially the air-turborocket engine analyzed in reference 8 where gasoline fuel was used with nitric acid as the oxidizer. The performance of the fuel-rich combustion cycle was not investigated in the present analysis.

Compressor pressure ratio and burner temperature. - The compressor pressure ratio possible with the present fuel expansion cycles considered (variations A, B, and C, fig. 15) is directly related to the work obtained by expanding the heated fuel and to the fuel flow rate. This pressure ratio, which is independent of the energy source used to heat the fuel, is shown in figure 16. For the operating conditions considered, the com-
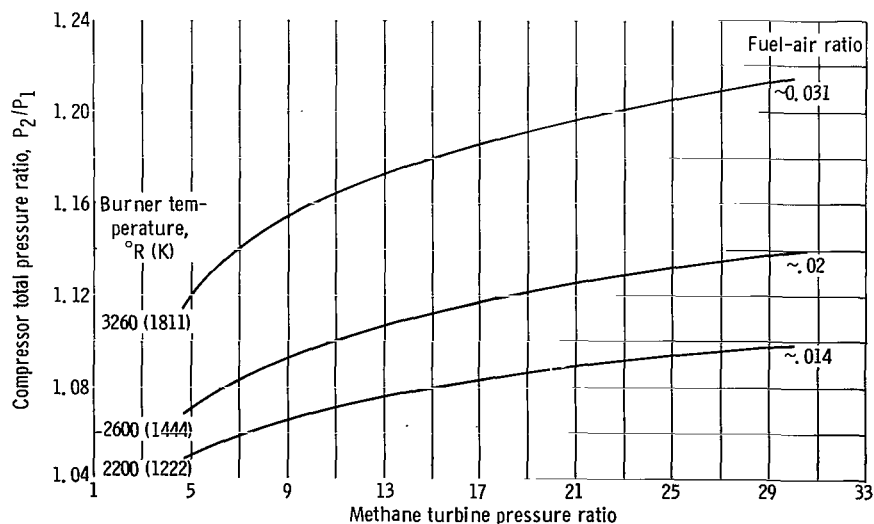

Figure 16. - Compressor pressure ratio of fuel expansion cycle. Methane turbine inlet temperature, 2000° R (1111 K); Mach 3.0 in stratosphere.

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pressor pressure ratios obtained are in the range 1.05 to 1.22. A methane turbine inlet temperature of 2000° R (1111 K) was arbitrarily selected and the turbine pressure ratio varied over the range 5 to 30. Burner total temperatures of 2200° R (1222 K), 2600° R (1444 K), and 3260° R (1811 K) were considered; and a combined combustor-heat exchanger pressure loss of 6.0 percent was assumed.

As shown by figure 16, an increase in compressor pressure ratio can be obtained by increasing either the burner temperature and, hence, the fuel flow rate or by increasing the methane turbine pressure ratio. Increasing the burner temperature raises the fuel air ratio f. Equation (16) shows that this will increase the work that is available to the compressor. A similar result is obtained by increasing the turbine pressure ratio and consequently the turbine specific work ΔH_t.

The performance obtained using the three variations of the fuel expansion cycle considered is shown in figure 17(a). Here specific impulse is plotted versus specific thrust for a range of burner temperatures from 2200° R (1222 K) to 3260° R (1811 K). For cycle variations A and C, the minimum burner temperature must be greater than the assumed methane turbine inlet temperature of 2000° R (1111 K), the left ends of the curves. A turbine expansion ratio of 30 was selected. This pressure ratio gives approximately the same maximum methane pressure that was used in the augmented preheat cycle with a turboexpander pressure ratio of 10.

When compared at equal values of specific thrust, the specific impulse of the three fuel expansion cycles is seen to be less than that of the reference turbojet. For example, at a specific thrust level of 40.1 seconds, the reference turbojet yields a specific impulse of 2790 seconds. At the same thrust level, fuel expansion cycle variations A, B, and C yield specific impulses of 2585, 2775, and 2665 seconds, respectively. While the specific impulse of cycle variation B is approximately equal to that of the reference turbojet, a heat source at 2000° R (1111 K) is not readily available. Therefore, a more realistic value of specific impulse is that of cycle variation C. Here the fuel is initially preheated to 1000° R (556 K) by an external source before passing to the tailpipe heat exchanger where its temperature is increased to 2000° R (1111 K). If higher specific thrust levels are desired, the tailpipe temperature must be increased with a consequent reduction in specific impulse.

It should be noted that the assumed pressure loss of 6.0 percent was used for cycle variations A, B, and C. Thus, the performance of cycle variation B which does not contain the tailpipe heat exchanger may be unduly penalized with respect to the other two cycles.

Effect of methane turbine inlet temperature. - As stated previously, a methane turbine inlet temperature of 2000° R (1111 K) was used in figures 16 and 17(a). The performance at reduced turbine inlet temperatures was also investigated, and the results are shown in figure 17(b). Here specific impulse is plotted against specific thrust for
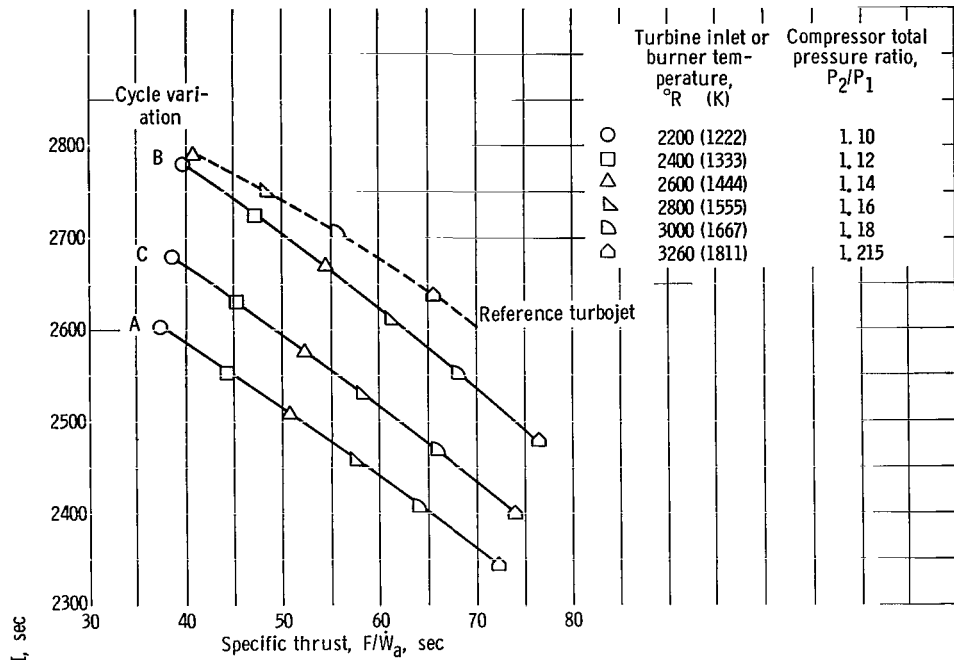
(a) Effect of burner temperature. Methane turbine-inlet temperature, 2000° R (1111 K).
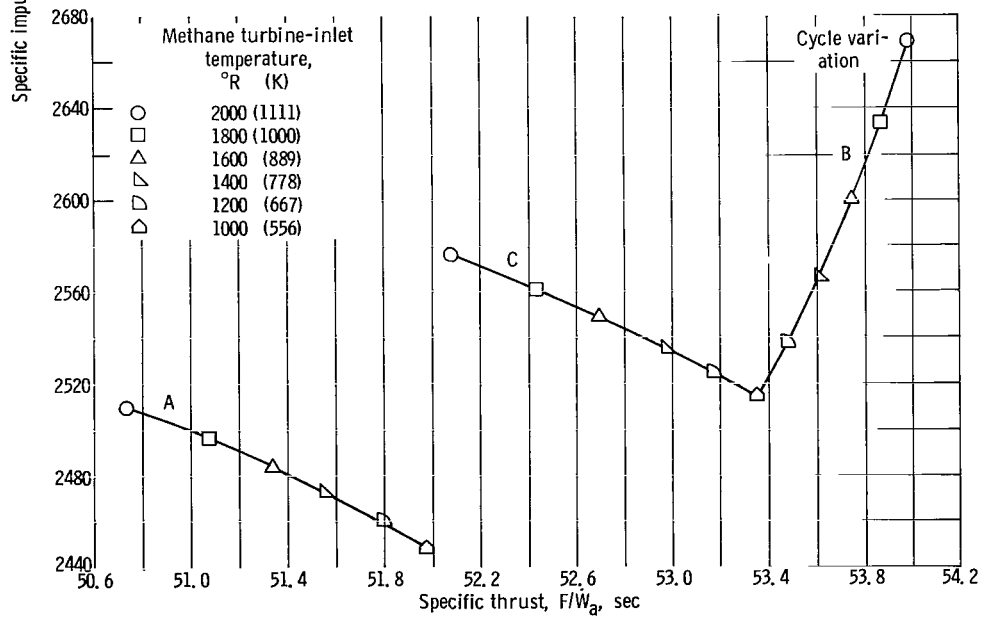
(b) Effect of methane turbine-inlet temperature. Burner temperature, 2600° R (1444 K).

Figure 17. - Performance of fuel expansion cycle. Methane turbine pressure ratio, 30; Mach 3.0 in stratosphere.

turbine inlet temperatures down to 1000° R (556 K). The turbine pressure ratio was maintained at 30 and the burner temperature fixed at 2600° R (1444 K).

At the turbine inlet temperature of 2000° R (1111 K), the performance obtained is the same as that shown in figure 17(a). It is seen that for the three cycle variations considered a decrease in turbine inlet temperature leads to a reduction in specific impulse. The specific thrust, however, can either increase or decrease depending upon the source of heat used to heat the fuel. For instance, if heat is obtained from the exhaust gases, as is the case with cycle variations A and C, a reduction in the temperature of the fuel into the turbine means that a lesser amount of energy is extracted from the exhaust. This will increase specific thrust. Note from figure (16) that a specific thrust increase is obtained for cycles A and C even though a lower turbine inlet temperature means a reduction in the compressor pressure ratio. For cycle variation B with the external heat source (and, hence, no heat extracted from the exhaust gases), a reduction in turbine inlet temperature will, as a result of the lower compressor pressure ratio, result in a decrease in specific thrust.

Comparison with turbojet cycle. - Figures 17(a) and (b) show that at the Mach 3.0 flight condition, the specific impulse of the fuel expansion cycles is less than that of the reference turbojet cycle at any given value of specific thrust. However, a meaningful comparison of this cycle with the reference turbojet cycle cannot be made by considering only specific impulse and specific thrust. For, unlike the previous cycles, the fuel expansion cycle differs considerably from the turbojet of figure 4. For example, the cruise compressor pressure ratio is reduced from 3.5 to less than 1.22. This implies that a lighter compressor could be used. In addition, the turbine no longer must be sized to pass the engine airflow. Also, the subsonic and transonic operating characteristics of the fuel expansion cycles may yield climb and acceleration profiles different than the other cycles considered.

SUMMARY OF RESULTS

The performance of a number of methane-fueled engine cycles was investigated in which the heat sink capacity of the fuel was employed for turbine cooling as well as for other purposes. Although the performance of one cycle is presented at sea-level static conditions, the primary cycle performance figures of merit are the Mach 3.0 cruise specific impulse, specific thrust, and the thrust per unit compressor frontal area. The performance of a turbojet cycle adapted to methane fuel was used as a reference for comparing the other cycles considered. No attempt was made to determine component weights for the various cycles investigated or to determine the weight of the total propulsion system. The following results were obtained:

(1) A gain in cruise specific impulse of approximately 1 percent was obtained at the reference turbine inlet temperature of 2600° R (1444 K) by eliminating compressor bleed air and using the fuel heat sink for direct fuel cooling of the turbine. In addition, the elimination of bleed air increased the specific thrust by approximately 1.2 percent.

(2) The performance of several cycles, which are modifications of the basic turbojet cycle, is presented below for a turbine inlet temperature of 2600° R (1444 K). Turbine cooling is provided with the turbine blade metal temperature fixed at 2200° R (1222 K). A maximum permissible methane temperature of 1680° R (934 K) is assumed.

Cycle	Percent change in performance from reference turbojet		
	I	$F/\dot{W}_a$	F/A_c
External fuel preheat (cruise)	3.2	-0.2	-0.2
Augmented external fuel preheat (cruise)	4.5	1.0	1.0
Compressor precooling (cruise, no recirculation)	.8	1.7	8.0
Compressor precooling (sea-level static, no recirculation)	.9	1.2	5.4
Compressor precooling (sea-level static with recirculation)	0	2.8	13.6

The performance of the compressor precooling cycle is shown for the minimum values of heat exchanger pressure loss associated with the heat transfer.

(3) When compared at equal values of specific thrust, the specific impulse obtained with three variations of the fuel expansion cycle was found to be less than that of the reference turbojet. However, this engine may weigh less than a turbojet engine of the same thrust. A maximum methane temperature of 2000° R (1111 K) was assumed.

CONCLUDING REMARKS

The gains in engine performance shown for the cycles considered are rather modest. This raises the question of whether a net gain in aircraft performance can be realized with these cycles when account is taken of the losses and weight associated with their practical implementation. The answer to this question depends on the mission requirements of the aircraft, the degree of airframe-engine integration obtained, and any side benefits these cycles may have.

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Several potential benefits of airframe-engine integration can be mentioned. For example, aerodynamic preheating of the fuel with airframe heat exchangers will tend to cool a portion of the aircraft structure. At Mach numbers of 3.0 and higher, this could yield significant savings in structural weight. In addition, if the cooling is done at the leading edge of the wing, the extent of the laminar boundary layer may be increased with a possible reduction in airplane friction drag.

The present analysis assumes that the power obtained by heating and expanding the fuel through the turboexpander is used to aid in driving the compressor. This power may be used more effectively to drive electrical, hydraulic, and other mechanisms that would otherwise require mechanical power extraction from the engine. With this scheme, the weight of the turboexpander and necessary plumbing may be compensated for by the elimination of power takeoff shafting and gearing.

Reduced noise may be a side benefit of the compressor precooling cycle. Engine exhaust noise is a strong function of exhaust velocity and a comparatively weak function of mass flow. Although compressor precooling increased the exhaust velocity slightly, the improved thrust per unit area obtained is largely due to increased mass flow per unit area. Therefore, for a fixed engine size, compressor precooling will yield thrust increases with a small noise penalty. In contrast, thrust increases obtained by afterburning will strongly affect exhaust velocity and will produce large increases in exhaust noise. Also, the introduction of a heat exchanger into the inlet duct may aid in reducing compressor noise.

Although the cycles discussed have been considered independently of one another, it may be advantageous to combine two or more in a single aircraft. For example, compressor precooling during takeoff could be combined with the cycle having augmented fuel heat addition at cruise.

The present cycle study investigates only part of the overall problem of making best use of the fuel characteristics. Although the gains in engine performance are generally modest, they may be significant when combined with related or additional benefits which can be realized in the overall airplane configuration.

Lewis Research Center,
    National Aeronautics and Space Administration,
        Cleveland, Ohio, April 23, 1968,
            126-15-02-02-22.

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